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EFFECT OF THRUSTER PULSE LENGTH ON THRUSTER-EXHAUST DAMAGE OF S13G WHITE THERMAL CONTROL COATINGS

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ABSTRACT

Rocket exhaust products which strike thermal control surfaces cause changes in solar absorptance (α_s) and thermal emittance (ϵ) of these surfaces.

A study was made of the effect of rocket pulse duration on exhaust damage to S13G white coatings. Two pulse lengths were used - 14 msec and 50 msec.

An MMH/N_2O_4 bipropellant 5-lb thrust rocket was fired into a simulated space environment with a vacuum of 10^{-5} torr, a liquid helium temperature enclosure, and solar radiation. The changes in solar absorptance and thermal emittance of S13 G white coatings due to rocket exhaust were made in-situ for total firing times of 58 seconds with 14 msec pulses and 223.7 sec with 50 msec pulses.

The solar absorptance of S13G increased 25 percent due to 223.7 sec of exposure to 50 msec pulses and the thermal emittance was unaffected. The ratio of α_S/ϵ therefore increased by 25 percent.

The short 14 msec pulse exhaust exposure caused between 40 and 70 percent increase in solar absorptance and a decrease of between 13 and 18 percent in thermal emittance. The corresponding increase in α_S/ϵ ratio was between 80 and 100 percent. Ultraviolet radiation was present in the short pulse test and may have contributed to the large damage of that test.

INTRODUCTION

Many spacecraft surfaces are exposed to the exhaust plumes of propulsion and attitude control thrusters. Some of these surfaces are commonly used as passive thermal control surfaces and are treated (by painting, polishing, etc.) to have specific optical properties (emittance, solar absorptance). The exhaust plume - spacecraft surface interaction may be such that changes in the optical properties of the surface occur. If such changes occur, the thermal balance of the spacecraft may be altered, resulting in intolerable spacecraft temperatures.

To study properly exhaust plume effects on thermal control coatings a thruster must be fired in a high-vacuum environment which must be maintained during the firing. Furthermore, properties of the coatings must be measured while the coatings remain in the environment. Three reasons dictate these requirements: (1) ambient pressures greater than 10^{-3} torr would cause the exhaust plume geometry to be different from plumes in space, altering the manner in which the plume impinges on nearby surfaces, (2) chemical reaction between the ambient gases and the exhaust products may change the type and amount of contaminant striking nearby surfaces, and (3) removal of the coatings for optical measurements would expose the contaminated coatings to atmospheric effects and perhaps alter the observable damage.

Previous plume damage studies reported in the literature[1-5] indicate the degree to which optical coatings are damaged by rocket exhaust plumes and show a wide range of degradation effects. But in most instances in-situ measurements were not made, or coatings were exposed to high ambient pressures during the rocket firings.

The Lewis Research Center's 6- by 8-foot liquid helium cooled space simulator[4] provides high cryopumping speeds so that very low ambient pressures can be maintained during a rocket firing. In addition, the simulator provides solar radiation so that in-situ measurements of optical properties of thermal control materials can be made. During the past several years studies of reaction control systems (RCS) thruster contamination effects have been made at Lewis Research Center.[4-8] These studies range from simulation requirements to determining effects of contamination on spacecraft materials. This paper reports on tests dealing with an additional factor in the thruster contamination of thermal control materials - that of the firing pulse lengths of the thruster. Other aspects of the contamination problem from these tests are being reported in companion papers at this meeting.[7-8]

Rocket engines are fired in two basic ways - (1) steady state and (2) pulse mode. During steady state firing, engine operation reaches equilibrium and only small quantities of unburned propellants are expelled in the exhaust. Pulse mode operation, however, is characterized by incomplete combustion and increases the amount of unburned fuel in the exhaust. Also residues from each preceeding pulse can be expelled by each succeeding pulse.

It is suspected that inefficient short fire time pulsing of a rocket engine results in more contamination and damage to surfaces than steady-state or long fire time pulsing. To investigate this possiblity, this paper reports on plume contamination effects on S13 G white thermal control coatings caused by two types of pulse-mode rocket operation. The pulse modes used were 14 msec pulses and 50 msec pulses. In addition, the tests discussed in this report included simulation of Solar thermal radiation and ultraviolet (UV) radiation. This radiation may interact with rocket exhaust contamination on surfaces and accelerate damage to those surfaces.

EXPERIMENTAL APPARATUS AND PROCEDURES

Test Facility

The rocket firings and optical measurements were made in the Lewis Research Center's 6- by 12-foot liquid helium cooled space simulater described in Ref. 4. A schematic of the space chamber and thruster installation is shown in Fig. 1.

The thruster being used is a 5-lb thrust version of the MOL reaction control system thruster (Marquardt Company). The engine uses as propellants monomethylhydrazine (MMH) and nitrogen tetroxide (N_2O_4). Both propellants were maintained between 13° and 20° C. The injector is a single doublet and the thruster is radiation cooled.

During rocket firings the test chamber pressure did not rise above 3×10^{-5} torr. Between rocket firings, a period of 7 to 10 minutes, the test chamber pressure fell to 6×10^{-7} torr as the helium cooled test chamber walls cryopumped the exhaust products.

Solar radiation is provided by a 23 kW carbon arc lamp which projects a collimated beam of radiation into the chamber.[9] The beam intensity can be controlled over the range of 0.5 - 0.7 solar constants (700 - 980 W/m^2) and is measured by radiometers installed in the plane of the samples. The radiometers are protected from plume contamination by electrical actuated covers which are only opened during the optical measurements.

Test Samples

The test samples are stainless steel type 302 discs 0.0127 cm thick and 2.46 cm in diameter. S13G coatings were put on the stainless steel substrates at Marshall Space Flight Center in accordance with that Center's specifications.*

Test samples are mounted in individual aluminum cups on three thin plastic pegs (figs. 2 and 3). Copper-constantan thermocouples are spot welded to the back center of the test sample and to the inside of the cup. The thermocouples are terminated at a plug at the bottom of the cup.

Test samples and mounting cups are secured in aluminum pallets horizontally mounted parallel to and about 10 cm below the thruster nozzle axis. Exact locations of the samples in the pallets are shown in Fig. 4. The samples are mounted so they are flush with the surface of the pallet. The flush mounting minimizes distrubances to the exhaust plume flow and simulates rocket flow over a spacecraft surface. In the experiments the pallets were maintained at temperatures in the range 230° to 260° K.

Test sample temperatures were similar in the two experiments and covered a range of 200° to 280° K. The wide range of test samples temperatures was due to the solar heating used for measurements as discussed later. During exposure to the rocket exhausts, test sample temperatures were usually in the range 200° to 220° K.

Test Procedures

Two tests were made that allow comparison of damage to surfaces from

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*S13G - potassium silicate treated zinc oxide in dimethylsiloxane: formulation per specification 10Mo1835; application per specification 10M01836; cured 40 hr at room temperature, 24 hr at 200° F in air circulating oven; thickness factor, 32 $in.^2$-mil/g.

long (50 msec) pulse mode and short (14 msec) pulse mode engine operation.

Performance of the engine was different in the two tests. Reference 4 discusses the build-up of engine thrust and propellant flow during early engine firing times. The engine's propellant valves have opening times of 8 msec and the ignition lag is 3 msec. Thus, for engine pulse lengths of 14 msec (short pulse test) the engine does not have enough combustion time to reach complete firing conditions. In fact, from Fig. 5, the combustion chamber pressure reaches only 40 percent of steady state values and the oxidizer/fuel ratio (O/F ratio) has wide excursions from the nominal steady state value of 1.5

Longer pulse lengths allow the engine to reach steady state operation. Pulse lengths on the order of 50 msec (long-pulse test) or longer are sufficient for the engine performance parameters to reach nominal steady state values.

An important factor in studying the contamination effect on the test samples is that the samples not be exposed to an ambient atmosphere. From the start of the tests until the end of the tests the test chamber was not opened to the atmosphere. During standby periods - overnights or weekends - the test chamber walls were ordinarily maintained at a temperature below 150° K and the test chamber pressure was held below 10^{-6} torr.

The foregoing refers in general to both tests. The specifics of each will now be considered.

Long Pulse Test: Over a span of several weeks the thruster was operated on 16 days for 6- to 8-hour periods. During these periods the thruster was fired in a pulse train mode consisting of 8 pulses of 50 msec duration with 100 msec offtime between pulses. The trains of 8 pulses were fired on a schedule that allowed 7 to 10 min between pulse trains. A total engine firing time of 223.7 sec was achieved.

During this long pulse tests, the test samples were exposed periodically to solar radiation during the tests when thermal measurements of α_s and ϵ were made. Overall, 15 hrs of solar exposure was experienced by the test samples.

Short Pulse Test: This test took place over a period of one and one-half weeks with engine firing occurring on 6 days. The engine was operated in a pulse mode consisting of 25 pulses of 14 msec duration, with 100 msec off time between pulses. The trains of 25 pulses were fired every 7 to 10 minutes. A total engine firing time of 75.5 sec was reached.

During the course of this short pulse test, the test samples were subjected to extended periods of solar radiation, alternating 60 min of solar radiation and 30 min of none. These periods simulated orbital solar heating. The solar radiation was at an intensity of 0.7 solar constants and a total radiation exposure time of 63 hr was achieved. Equivalent sun-hours is defined as intensity multiplied by exposure time so that the test samples experienced 44.1 equivalent sun hours exposure to solar radiation.

Supplemental ultraviolet radiation was also provided by a deuterium lamp source which gave radiation over the wavelength range 2000° to 3000° A. The lamp was mounted within the test chamber and added to the UV radiation of the carbon arc solar simulator. The UV lamp was 50 cm above the test pallet and except for a few hours at the start of the test it was on continuously. The deuterium lamp provided 0.033 solar constants of UV radiation over the range 2000° to 3000° A. A total exposure time of 221 hr was reached, or 7.3 equivalent sun hours of UV exposure.

α_s and ϵ Measurements

In-situ thermal/optical measurements were made by heating the samples with the solar beam, allowing them to cool and then calculating solar absorptance (α_s) and thermal emittance (ϵ) from the heating and cooling rates. Similar techniques have been previously used by others and ourselves.[7,10,11]

The thermal model used includes solar heating, sample-mounting cup heat exchange, sample radiation, and engine-shroud heating. Equation (1) expresses the basic thermal model.

$$A\left(\rho_b C_{p_b} d_b + \rho_c C_{p_c} d_c\right)\frac{dT_c}{dt} = -\epsilon\sigma A T_c^4 - K_r\left(T_c^4 - T_m^4\right) + \alpha_s \varphi A + S T_{shrd}^4 \qquad (1)$$

where

A	test surface area
c_p	specific heat
d	thickness
K_r	radiation transfer coefficient between mounting cup and test surface
S	radiation transfer coefficient between test surface and thruster package (shroud)
T_m	coating temperature
T_{shrd}	thruster package (shroud) temperature
t	time
α_s	solar absorptance
ϵ	thermal emittance
ρ	density
σ	Stefan-Boltzman constant

φ solar radiation flux

Subscripts:

b substrate

c coating

In brief, Eq. (1) is used in the following way. All the parameters except α_s and ϵ in Eq. (1) are known. Test samples are weighed before and after painting and specific heats of the coatings and substrates were taken from Ref. 12. Temperatures and solar intensity are measured as a function of time and heating and cooling rates are calculated. ϵ and α_s can then be determined from Eq. (1) by analytical techniques discussed in detail in Ref. 5.

RESULTS AND DISCUSSION

The results of the two tests are shown in table I and Figs. 6 and 7.

Over the course of the long pulse test (fig. 6) and solar absorptance slowly increased so that after 223.7 sec of engine firing the α_s of sample 28 had increased from 0.20 to 0.25. The thermal emittance of sample 28 was unaffected.

The short pulse test (fig. 7) showed a much more rapid increase in α_s so that after only 58 sec of engine firing time α_s of samples 50 and 51 increased 45 and 73 percent. In addition, in contrast to the long pulse test, ϵ of samples 50 and 51 decreased about 18 and 13 percent, respectively.

The data indicate that the 14 msec rocket pulsing environment was much more damaging to the S13G coatings than was the longer 50 msec pulse environment. This is in agreement with the results of Ref. 11 which reports damage to Quartz transmitting elements and mass deposition measurements for the same two tests.

The short pulse and long pulse tests differed in ways other than the pulse durations. The short pulse test included more solar thermal and UV radiation. This may, to some extent, account for the accelerated damage from the short pulse test. But the mass deposition measurements of Ref. 11 indicate that the less efficient engine operation in the short pulse mode (discussed earlier) is most likely the cause of greater contamination and damage.

Test samples in the long pulse test were sugjected to extensive solar heating at the end of the test. No effect was noted on the optical properties of the S13G samples due to the extended solar heating.

Ultraviolet radiation damage has been noted on white paints. References 13 and 14 report this type of degradation of white paints on spacecraft. Those studies showed that 10 to 15 ESH of exposure to solar radiation are needed for changes in α_s to be noticed. Thus, the 7 ESH of

UV radiation in the short pulse test would not be expected to damage the coatings of that test. The studies mentioned[13,14] were on actual spacecraft in space but with no rocket engines in the environment. Thus there were not exhaust products available to deposit on the test surfaces. So it is still possible that there is a synergism between the rocket exhaust products and UV radiation. This possibility is being investigated at the Lewis Research Center.

SUMMARY

In-situ measurements of solar absorptance (α_s) and thermal emittance (ϵ) were made of S13G white paint exposed to rocket exhaust plumes. Two rocket pulse modes were used - a long pulse mode (50 msec fire time) and a very short pulse mode (14 msec fire time).

After 223.7 sec of exposure to the long pulse exhaust, α_s increased 25 percent and ϵ was unaffected. The ratio α_s/ϵ increased by 25 percent. During the short pulse exhaust test, only 58 sec of exposure caused from 45 to 73 percent increase in α_s of the S13G paint and from 13 to 18 percent decrease in thermal emittance ϵ. The ratio α_s/ϵ increased between 80 and 100 percent.

The conclusion drawn is that very short pulse mode operation of a rocket engine with UV radiation degraded the optical properties of S13G paint considerably more than a longer pulse mode operation of a rocket engine without UV radiation.

ACKNOWLEDGEMENT

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TABLE I. - S13G

Pulse train type	Total engine firing time, sec	Sample number	Initial properties		Final properties	
			α_s	ϵ	α_s	ϵ
8-50 msec fire time, 100 msec off	223.7	28	0.20	0.83	0.25	0.83
25-14 msec fire time, 100 msec off	*58	50	.16	.83	.23	.67
		51	.14	.85	.24	.74

*This time is less than the total 75.5 sec of actual engine firing time reached in the short pulse test. At the end of this test the data recording equipment malfunctioned so that beyond 58 sec no further useful optical property data was recorded.

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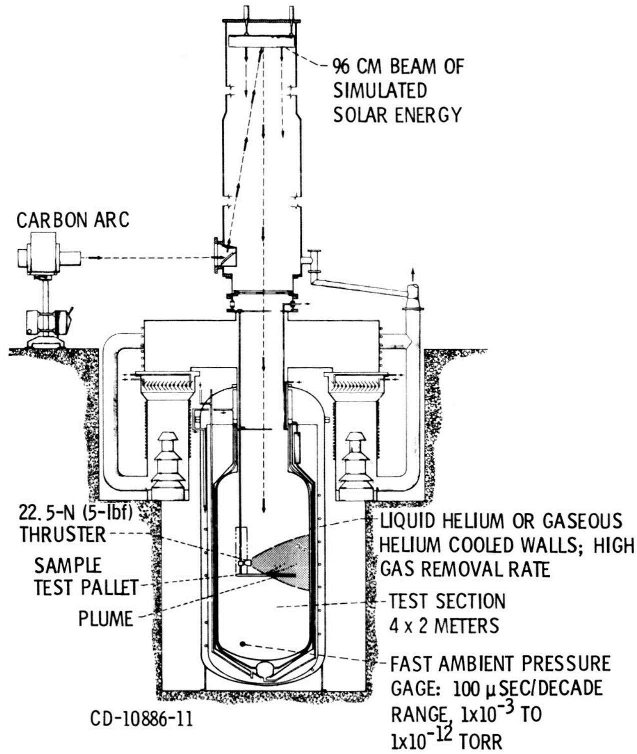

Figure 1. - Liquid helium cooled space simulator and thruster installation.

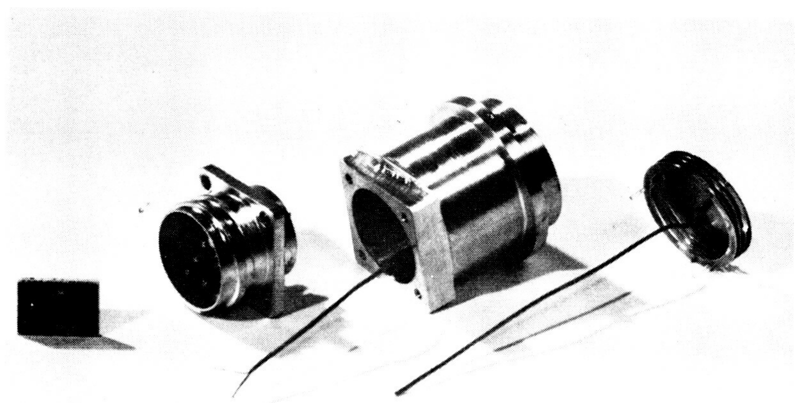

Fig. 2 Sample and mounting cup assembly.

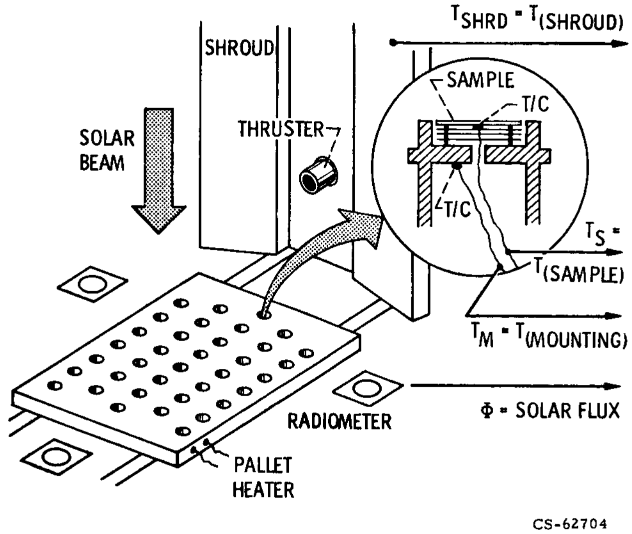

Figure 3. - Sample arrangement.

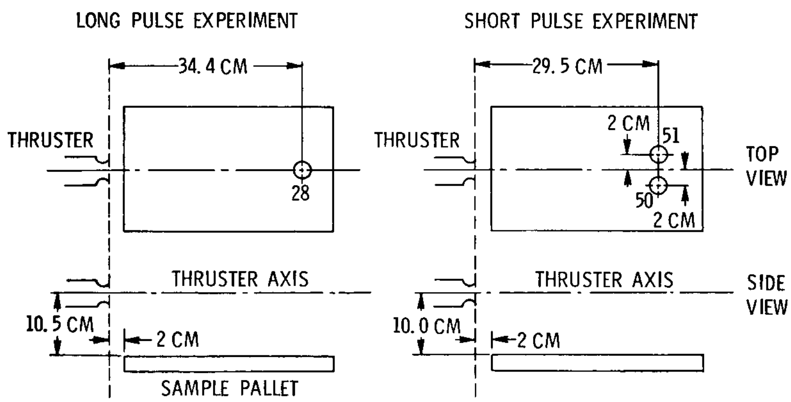

Figure 4. - Sample locations.

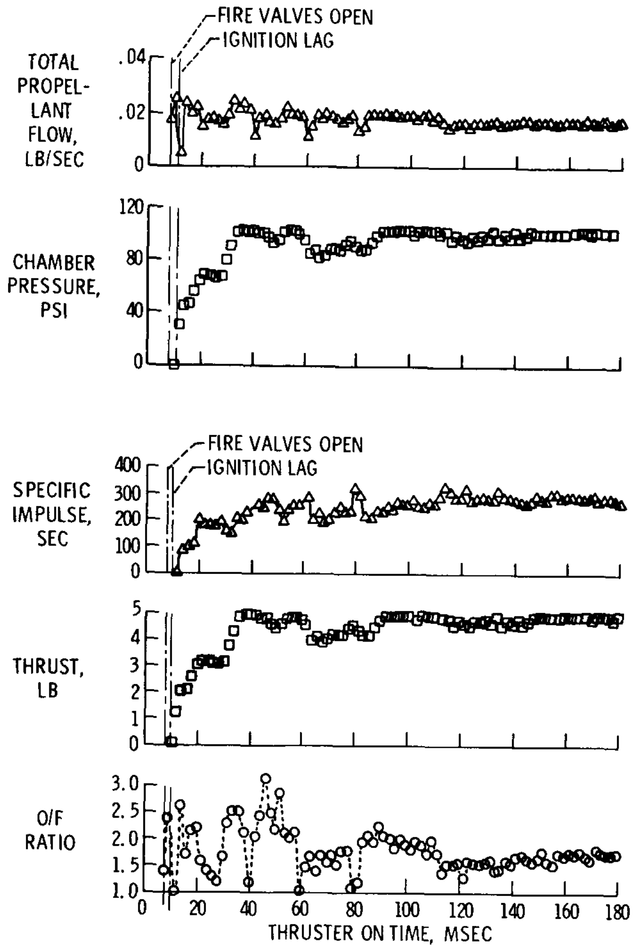

Figure 5. - RCS thruster performance.

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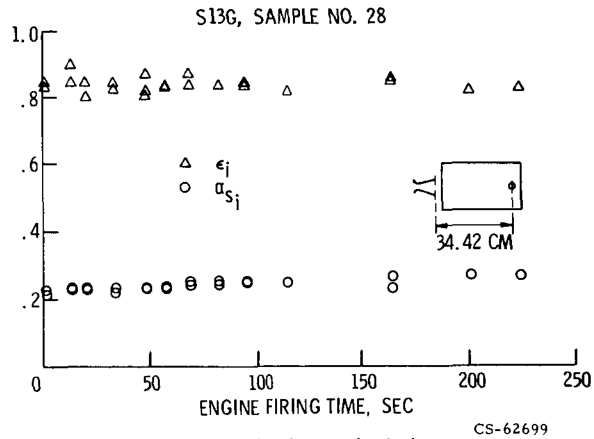

Figure 6. - Long pulse test.

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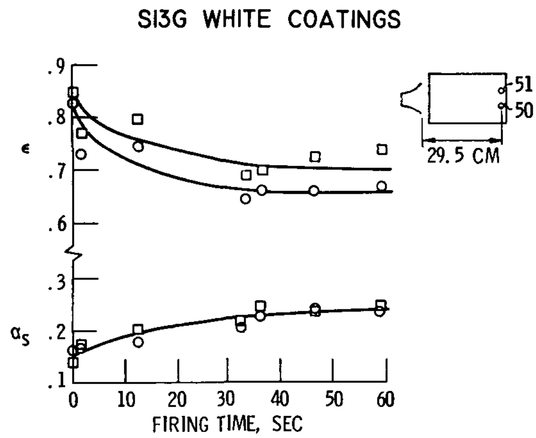

Figure 7. - Short pulse test results.

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